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Abstract

The SERT II spacecraft, launched in 1970, has been maintained in an operational, but intermittently active status since 1971. Periodic thruster status has been reported while waiting for normal orbit precession to return the spacecraft to continuous sunlight in 1979. Now, the thruster has been operated for 600 hours in the first quarter of 1979. Thruster startup and operation in 1979 is unchanged after 9 years in space. The ion thruster was gimbaled and used to maintain spin stabilization of the spacecraft. Minor components of the spacecraft have failed, but have not interfered with the functional status of the spacecraft.

Introduction

The Space Electric Rocket Test II (SERT II) spacecraft was launched in February 1970 with a goal of demonstrating long-term operation of an ion thruster system in space. Thruster 1 was operated for 5-1/2 months and then thruster 2 was operated for 3 months.[1] Each thruster operation was terminated by a high-voltage grid short. In 1973, new goals were added for the still functional SERT II spacecraft: (1) demonstration of thruster restartability after long space storage; (2) study of factors limiting thruster restarting; (3) demonstration of the space lifetime of thruster system components, such as, propellant feed systems, closed-loop control systems, insulator shields, and power processor units; (4) measurement of main solar array degradation after years of space exposure; and (5) verification of the compatibility of the spacecraft systems with sustained thruster efflux. Progress towards these goals was made in 1973 with the demonstration of over 112 restarts of each thruster.[2]

In 1974, the high-voltage short of thruster 2 was cleared and full operation of thruster system 2 was restored when the spacecraft was placed into a spin-stabilized mode. In the 1974 to 1978 period, testing to the 1973 goals was continued, and the additional goal of periodic full operation of thruster 2 was added.[3,4] Steady-state operation for more than 1 hour per orbit during 1973 to 1978 was prohibited by loss of solar array power due to a shadow period occurring each orbit revolution.

Natural precession of the SERT II orbit has now brought the spacecraft back into long periods of continuous sunlight (1/79 to 4/79 and 8/79 to 8/80) and steady-state testing is feasible. The 1979 test goals include all those of 1973 and, in addition, the following: (1) to demonstrate the steady-state operation of an ion thruster system 9 years after launch, (2) to measure and compare the performance of this aged thruster with that of a new thruster, and (3) long term operation of the ion thruster system to determine its life-limiting factor. In meeting the 1979 goals, the ion thruster was used to increase the spacecraft spin rate to maintain spacecraft attitude stability. (Prior to 1979, spin rate was maintained by cold gas jets.) At the same time, a direct thrust measurement was obtained for the ion thruster.

This paper contains the results of the January to April, 1979 steady-state testing of thruster 2 for 599 hours, the continued restart tests of thruster 1 (its internal short remains), and a report on the general status of all spacecraft systems including the main solar array. The results of neutralizer cross-coupling and bias experiments concurrently conducted in 1979 are contained in a companion paper.[5]

Apparatus and Procedure

SERT II Spacecraft

Figure 1(a) is an artist's drawing of the SERT II spacecraft. A detailed description may be found in the literature.[1,6] The spacecraft was launched in 1970 into a 1000-km, circular near-polar (99° inclination) sun synchronous orbit. Reference 7 describes the spacecraft gravity gradient orientation from launch to 1973, and the need to spin stabilize after 1973. Natural orbit precession resulted in an initial continuous sun period, lasting until the end of 1971. Partial shadowing of the orbit occurred from 1972 through 1978. Continuous sun again occurred from January 9, 1979 to April 11, 1979, and is predicted at this writing (June 1979) to occur from August 1979 to August 1980. The spacecraft battery reached end of life before 1973, and the only spacecraft power source now is the solar array.

The basic spacecraft structure is an Agena vehicle with a fold-out solar array mounted off the aft rack and two ion thrusters mounted on a forward deck as shown in figure 1(b). Although never designed to be spin stabilized, the spacecraft has been spun about an axis perpendicular to the plane of the solar arrays. Presently the spin axis is approximately perpendicular to the orbit plane. A mechanical analysis of the solar array structure indicates that buckling of solar array supports may cause uncontrolled distortion of array panels at a spin rate in excess of 6 to 9 rpm. Therefore, the spin rate of the spacecraft has been limited to 6 rpm or less. Stability has been demonstrated at spin rates of 0.3 rpm. Periodic spinup is required to make up for natural damping forces which result in a spin rate loss in the range of 1 to 6×10^{-3} rpm/day. The cause of damping has not been analytically modeled, but the despin rate appears to be a function of the spin magnitude and the spin axis direction.

The attitude of the spacecraft has been determined by the roll-offset of the Agena horizon scanners. The spin rate has been measured by both the pitch-horizon crossing of the scanner and by the automatic gain control (AGC) signal level variations from tracking station receivers.

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The performance and degradation of the solar array, as well as a status of spacecraft systems, is presented in the Results and Discussion section.

SERT II Ion Thruster Systems

The two identical flight thruster systems have been described in the literature[1] and only that description bearing on the results of this report will be breifly repeated. Figure 2 is a cutaway drawing of one thruster. Each mercury ion bombardment thruster is 15 cm in diameter, operates at a nominal 1 kW and produces 28 mN (6.3 mlb) of thrust at 250 mA beam of 4200 second specific impulse. In addition to the 250 mA beam set point there are set points available for operation at 200 and 85 mA beam. The screen voltage is unregulated and is proportional to solar array voltage. The nominal 4200 second specific impulse is produced by 3000-volt screen voltage at 60-volt solar array voltage (See solar array Appendix for circuit diagram.) The screen and accelerator supplies are protected against current overloads and briefly turn off (0.1 sec) if the screen (beam) current exceeds 300 mA or if the accelerator current exceeds 50 mA. This amount of off time (0.1 sec) is usually enough for an arc to clear itself. The power processor unit (called "power conditioner" on SERT II) was designed to operate with a minimum input voltage of 48 volts. Input voltages below 48 volts cause automatic turn off of the power processor unit and requires ground commands to restart.

Results and Discussion

Thruster 2 Operation

Thruster 2 operated for 599 hours during the period of January 22 to April 5, 1979. Table 1 lists each of the nine test segments and twelve starts that were made in accumulating the total of 599 hours. The thruster was operated at a nominal 85 mA beam current except for 26 hours, during which time it was operated at a beam current of 200 mA. Operation was attempted several times at 250 mA beam current, but insufficient power was available from the naturally degraded solar arrays to permit this operation. Table 2 lists representative values for thruster 2 operating parameters during preheat, propellant and beam operation for periods from 1970 to 1979.

As can be seen from Table 2, thruster 2 electrical performance is nearly identical after a 9-year time span for all operating ranges tested. Minor differences fall within telemetry uncertainty or are the results of known set point changes.

The gimbals on thruster 2 were positioned to offset the thrust vector from the spacecraft center of gravity and produce a torque which increased the spacecraft span rate. Knowing the gimbal position and measuring the spin rate increase enabled a value of thrust to be determined. Appendix A derives an equation for this calculation and presents a discussion of the gimbal system.

Figure 3 is a plot of the spacecraft spin rate during the first part of 1979. The horizontal portions of the data plot are periods when the ion thruster was off. A slight loss of spain rate during these periods was due to natural damping forces. Thrusting began on day 22 and was concluded on day 95. When the thruster was on, the spin rate shows a positive slope. This slope is proportional to the thrust magnitude and the sine of the gimbal angle (appendix A). Sections of figure 3 with nearly vertical decreasing spin rate values correspond to the use of on-board gas jets to reduce the spin rate. This was done to keep the spin rate below 6 rpm and prevent structural damage to the solar array.

The slope of figure 3 for run 5 resulted in a computed thrust value of 10.0 millinewtons (mN) at 85 mA beam current. This value was within telemetry uncertainly (data accuracy) of the thrust value calculated from electrical parameters of 9.8 mN. The thrust value measured in 1979, when compared to the 1970 250-mA beam current data, agrees within data accuracy of the thrust measurements (±3%). Table 1 gives individual values of beam current and voltage, test segment hours, and gimbal angle for each of the nine thrust-producing test segments. The conclusion of the thruster performance data of figure 3 and Table 1, is that there has been no change in thrust level or efficiency of the thruster system in the 9 years since launch.

At the time of the SERT II launch, 1970, the life-limiting factor of the thruster system was believed to be erosion of the main cathode orifice.[1] Such erosion enlarges the orifice and permits more propellant to flow through the cathode. This increased flow tends to increase the discharge current. A life limit occurs when the discharge supply current limit is reached. In-flight cathode erosion was less than experienced in ground tests and was never a life-limit factor in flight thruster operations. This was one of the basic results fround in 1970.[1] Operation of thruster 2 for 599 hours in early 1979, resulted in no change of the main discharge current. Thus, the earlier result of low, in-flight, cathode orifice erosion is also true for thruster operation at the throttled level of 85 mA beam current.

Thruster 2 Starting

Thruster 2 was started successfully 12 times during the first part of 1979 and never failed to start when attempted. The starting times and heater values are listed in Table 3 for the entire history of the thruster from 1969 to 1979. As can be seen in Table 3, the starting times in 1979 are nearly identical with those of 1970. The longer starting times between 1973 to 1977 are due to cooler initial thruster temperatures caused by partial orbit eclipses during that time period.

In addition to easy starting, the heater electrical values show little or no deterioration after 230 restarts and 2930 hours of vaporizer/cathode operation. For the first start attempt of 1979, start number 219, the traditional starting procedure was followed. That procedure consisted of one orbit 106 min) of preheat followed by one orbit of propellant (discharge only) before turning on the H.V. beam. Based on the quick starts of 1973 to 1978 periods and on a practical desire to reduce starting cycle times, a shortened starting procedure was adopted for the balance of the 1979 starts. This shortened starting procedure called for 20 minutes of preheat (to warm up propellant manifolds) followed by 5 minutes of propellant (to stabilize the vaporizer control loop) before turning on the H.V. beam.

The SERT II thruster system was designed to test the steady-state lifetime capability of an ion thruster. Quick starting was not a requirement and the thermal response of the propellant feed system is slow due to relatively heavy metal parts and small heaters. Nevertheless, the modified starting procedure comes close to that desired (15 min) for present cyclic ion thruster systems, and indicates that with proper design this startup time should be met. Minimum startup time is desired for cyclic ion thruster systems to maximize propellant efficiency, and minimize electrical energy requirements.

The only starting difficulty occurred during run 2 (Table 1) when the discharges lit properly, but due to a ground command problem, the H.V. beam-on command was sent at a non-optimum time, when the vaporizers were unstable. The resulting beam current produced by the thruster exceeded the power capability of the solar array. A power conditioning undervoltage turnoff then occurred automatically when the array voltage dropped below 48 volts.

Thruster 2 Arcing

Table 1 gives the number of high-voltage recycles or arcing that occurred during the 599 hours of early 1979 operation. A total of only eight arcs was logged; this rate is considerably lower than that experienced in 1970. In 1970, the mean period between arcs was 6 hours, with several periods of 50 hours and one of 150 hours with no arcing. The probable cause of reduced arcing in 1979 was operation at lower beam currents (85 versus 250) with a corresponding lower buildup of condensed sputter metal on the accelerator grids to initiate arcs.

Runs 1, 3(b), 5, and 9 were terminated when a high-voltage arc occurred and the power conditioner experienced an undervoltage shutdown. When an arc occurs accelerator current reaches its trip level of 50 mA and initiates an arc recycle sequence. The main vaporizer, screen (beam), and accelerator supplies are turned off for 0.1 second. For a single arc event the thermal time constant of the vaporizer is long enough to ignore the effect of the vaporizer supply turnoff. With the high-voltage supplies off (V5 and V6), few ions are being extracted from the discharge chamber. When the high voltages are reestablished at the end of the "blink-off" cycle an overshoot of beam current occurs above the desired beam setpoint. This overshoot of beam current returns to desired levels within 10 milliseconds as the excess ions are depleted. To prevent the normal beam overshoot at turnon from initiating another recycle the trip is not enabled until 0.01 second after high voltages are turned on, this allows sufficient time for the overshoot condition to return to below trip levels. The beam trip set point is 300 mA. The degraded solar array is incapable of supplying sufficient steady-state power to meet this demand and will cause the solar array voltage to drop below 48 volts tripoff point of the power conditioner. The authors believe that runs 1, 3(b), 5, and 9 were shut down by overpower demand following an arc and subsequent beam overshoot upon recycling of the H.V. power supplies. If the beam overshoot is not too high, the power supply input capacitance may supply enough temporary power to avoid overloading the solar array.

On runs 1 and 8, Table 1, the arc clearing circuit cycled without causing undervoltage shutdown. On runs 4, 6, and 7, the power conditioner went through an undervoltage shutdown due to intentional test procedures which placed an excessive load on the solar array. For tests 8, 10, and 12, the test was terminated by planned ground command.

Test 11 was unique. One objective of test 11 was a quick H.V. turnon before discharge ignition to measure H.V. leakage current with no beam or discharge on. No leakage was measured, thus confirming the design of thruster H.V. insulators. However, after 1-1/2 minutes of this condition, a grid short appeared. Its characteristics were consistent with and similar to those grid shorts in 1970 that terminated beam extraction. After continuous overload recycles for 2 minutes an overload integrator shut the system down. Presumably a web of grid material had shorted across the grids. A second attempt to turn on H.V. resulted in similar arcing, and this state was permitted to continue (by disabling the overload integrator) for 1 hour with no change.

In an attempt to clear the short, the thruster was restarted in a cold state, where thermal differential contraction might result in breaking off of the shorted web fragment. The attempt was successful and when H.V. was applied to a cold thruster, the short was cleared. Subsequently, thruster 2 ran for 49 hours and was commanded off with no problem. It was turned off because the spacecraft orbit was approaching the calendar time of partial orbit shadowing which was the planned end of this test period.

Thruster 1 Restarts

Thruster 1 was restarted 13 times between January 17 and May 22, 1979. Representing starting times and heater values are listed in Table 3. The same starting results were found for thruster 1 as for thruster 2, that is, similar starting times as a function of the initial temperature state of the thruster, and little or no change in heater resistances. On several occasions the H.V. was applied to thruster 1 with no change in the existing grid short. The discharges of thruster 1 were turned on for 172 hours during the above period to supply a space plasma for performing cross-neutralizing experiments reported in a companion paper.[5]

Neutralizer Propellant Tanks

Each neutralizer propellant tank was equipped with a transducer to measure the gas (80% N_2, 20% Kr) pressure in the blowdown volume. Knowing the gas volume and temperature, and measuring the pressure rate of change, enables a calculation to be made of the neutralizer mercury propellant flow rate. Figure 4 is a plot of the pressure change with vaporizer operating hours for the neutralizer of system 2. The pressure reading was constant to 1570 hours because radiated thruster heat caused the gas pressure rise beyond the transducer maximum range. At 1570 hours the pressure indicated a drop of one telemetry count. Thereafter there was a drop of one count approximately every 110 hours until the grid short occurred at 2980 hours in 1970. From day 78, 1971 to day 22, 1979 approximately 50 hours of vaporizer flow occurred, but the pressure decayed 4.0 N/cm^2 (5.8 psia). The excess decay, 4.0 N/cm^2 was assumed to be a gas leak to space. The pressure lost over 8 years corresponds to a low leak rate of 1.7×10^{-4} cm^3/hr (STP).

In 1979, thruster neutralizer vaporizer was operated for 602 hours and the pressure decayed from 37.2 to 33.0 N/cm^2. The calculated flow rate for 1970 was 27 mA average equivalent flow of mercury, and for 1979 was 37 mA. The higher flow rate measured in 1979 is in agreement with laboratory tests in which higher neutralizer flow rates were required at the throttled beam (1979) as compared to full beam (1970). The estimated pressure at which the tank will be empty is 27.2 N/cm^2 and will occur after an additional operating time of 1100±200 hours. Approximately 35 percent of the original mercury propellant remains. The leak rate of gas is minor and will not be a factor in limiting the flow of mercury. A surplus of gas of 22.2 N/cm^2 will exist if the propellant is exhausted in 1980 or earlier. The integrity of the design is presently verified for a period of over 9 years in space.

The plot of neutralizer propellant tank pressure for system 1 is similar to that of figure 4 and is not presented. The apparent leak rate of this tank for the same period of 1971 to 1979 is 1.5×10^{-4} cm^3/hr, and is also of no concern. Approximately 35 percent of the original mercury remains in the tank.

Solar Array Performance

Performance of the SERT II solar array has been reported in the past.[4,9] The data obtained during continuous thruster operation during the first quarter of 1979 has provided an update of the array status. Because operation was continuous, and not intermittent as during previous testing periods (1974 through 1978), a higher degree of accuracy exists for the current data. Figure 6 shows a plot of the projected radiation degradation and data points observed during this and previous operating periods. As shown, the degradation is following the projected trend, but is consistently a few (3 to 5) percent less than projected. Array degradation being less than expected indicates no significant contamination from spacecraft sources including ion thruster operation.

Additional detail on the solar array characteristics can be found in appendix B.

Spacecraft Systems Status

The extended operation of the SERT II spacecraft has been possible because no critical spacecraft failures have occurred with an orbit operation now in its 10th year. A brief summary of current subsystem status follows:

Thermal - Passive, except for P/C heaters, still provides component temperatures within original limits.

Power - Battery failed only after it exceeded its lifetime, all other power system components (inverters, regulators...) performing normally.

Data - One subcommutator has failed resulting in the loss of 60 out of the 1200 data channels. Several of the channels affected were redundant on other telemetry channels. One of two tape recorders has failed.

Attitude Control System - Four control moment gyros (CMG's) and horizon scanners fully functional.

Backup Attitude Control System - This cold gas system is fully functional with about 50 percent of the gas remaining.

Secondary Experiments

- Beam probes - one fully functional, the second is partially functional
- Reflector erosion experiment - fully functional
- Miniature electrostatic accelerometer - not functional
- RFI experiment - not functional

Selected Component Operating Time

The operational hours on selected components are given in the table below. The total orbital life of SERT II satellite to date is approximately 83,000 hours. A history of SERT II operations is given in figure 6.

Operational, hr	Component
64,000	Command system, switching mode regulators, and command receivers
26,000	Data system
25,000	Battery charger (turned off when battery failed)
25,000	Transmitter 1
21,000	Main inverter
20,500	CMG's 3 and 4
14,000	Transmitter 2
10,000	Standby inverter
6,000	CMG's 1 and 2
4,100	Power conditioner 1
3,000	Power conditioner 2
650	Horizon scanners

Conclusions

The SERT II spacecraft orbit has precessed sufficiently, in 9 years since launch, that the spacecraft again is in continuous sunlight. This sunlight provides continuous solar array power for thruster operation. During the first quarter of 1979, thruster 2 was operated for nearly 600 hours at 85 mA beam current. The performance of the thruster was found to be unchanged in the 9 years since launch. Restarting was also normal during 12 restarts. The thrust of the ion thruster was used to spinup the spacecraft to maintain spacecraft attitude control, and in doing so, an additional measurement of thrust was achieved. This thrust measurement showed no change from the values obtained in 1970. The solar array degradation was measured and found to be slightly less than predicted, thus indicating no additional degradation due to space-

craft sources or ion thruster efflux. The spacecraft remains functional and has only experienced minor failures of two secondary experiments, a battery, one sub commutator (60 of 1200 data channels) in the telemetry system, and the loss of one of two tape recorders. Approximately 35 percent of the neutralizer propellant remains and additional testing of the thruster systems is planned when the spacecraft regains continuous sunlight in the August 1979 to August 1980 period.

Appendix A - Spacecraft Angular Acceleration and Thruster Gimbal System

Figure 7 is a line drawing of the SERT II spacecraft. The spacecraft is spin-stabilized about an axis perpendicular to the paper and running through the center of gravity. Both ion thrusters are located such that their thrust vector nominally passes through the spacecraft center of gravity. The thrust vector can be gimballed, ±10° about the nominal thrust vector. Referring to figure 7, the clockwise angular acceleration about the center of gravity is given by the equation:

$$\text{Angular acceleration} = \frac{\text{Force} \cdot \text{distance} \cdot \sin\theta}{\text{Spin axis moment of inertia}}$$

where

Force = thrust of ion thruster

$$= 2.04\times10^{-3} \cdot (I_5) \cdot \sqrt{V_5 + 20} \text{ newtons}$$

I_5 is beam current, amperes

V_5 is screen (beam) voltage, volts

(beam divergence and multiple ion loss is less than 1% and is neglected)

Distance = distance from thruster gimbal mount to spacecraft center of gravity, 3.04 meters

θ = gimbal angle, degrees

Moment of inertia of spacecraft about spin axis = 10,970 kg-meters^2

For an 85-mA beam current at 3000 volts, the thrust is 9.5×10^{-3} newtons. Using a gimbal angle of +10° (as shown in fig. 7) results in a calculated angular acceleration of 4.6×10^{-7} radians/sec^2 or 0.38 rpm/day. The natural decay rate of the spacecraft varies with spin rate and is $(2 \text{ to } 8)\times10^{-3}$ rpm/day. This decay tends to reduce the net spin rate increase and is subtracted as a correction to the calculated value of angular acceleration.

Thruster Gimbal System

Each ion thruster is mounted on two independently controlled gimbal actuators to allow vectoring of the thruster. One gimbal drive (shown in fig. 8) vectors the thrust angle as shown in figure 7 and varies the magnitude and sign of the spacecraft angular acceleration. The other gimbal drive varies the thrust vector in the cross direction. This gimbal is normally positioned in the center position, but even if it were not, its position would have a small effect on the average angular acceleration. No telemetry readout of gimbal position is provided. Position is determined or set by commanding to either limit of travel and then timing the travel back to the desired position. Additional details of the gimbal system may be found in reference 8.

During gimbal operation in the first quarter of 1979, anomalous gimbal movement occurred. The motion appeared normal in the direction to increase θ (as shown in fig. 7), but motion in the opposite direction appeared to be retarded. This motion was deduced by analysis of the data from spacecraft spinup by the ion thruster. The calculated angular acceleration equation above has two unknowns, thrust magnitude and gimbal position. By driving the gimbal in the full positive θ direction, (run 10, Table 1) its position was known and the measured angular acceleration was used to calculate thrust. The thrust value so measured was in exact agreement with the thrust values calculated from electrical measurements and also those thrust values measured in 1970. For other gimbal positions, the thrust value was assumed constant and the gimbal position was calculated. These calculated gimbal positions are listed in Table 1. A gimbal θ position of ±5.5° appeared to be the limit of gimbal travel in the direction towards negative θ values. If the gimbal travel had gone to negative θ values, the ion thruster could have been used to despin (to avoid mechanical damage to solar array) the spacecraft instead of the cold gas system.

Appendix B - Solar Array Description

The SERT II solar array consits of 33,300 (N on P) 2×2 cm solar cells. The solar cells are 12-mil-thick soldered cells, with a base resistivity of 1 to 3 ohm-centimeters. Solar cell coverslides are 20 mil fused silica. The performance data presented here is derived from flight data on the 25,840 cells that make up the "thruster" section of the solar array. The thruster section of the solar array is configured as 38 parallel connected sections of 2 series connected panels each. The 76 panels (2×38) each consist of 68 series connected submodules. Each submodule is made up of 5 parallel connected solar cells. Submodule construction, panel layout, and solar wing configuration is shown in figure 9. For reference, the electrical configuration of the SERT II solar array/power system is shown in figure 10.

Flight data (array voltage, current, and temperatures) obtained during various load configuration has been reduced and normalized to a common baseline. Solar array characteristics for beginning of life (BOL) in 1970, and its status as observed in the first quarter of 1979 are presented in figure 6.

In generating the data set to plot the present array characteristics the best probable sun angle was assumed in each case. Typical sun angle uncertainty is ±5 degrees or about ±4 percent of the solar flux intensity. This gives a somewhat pessimistic result for calculated solar array parameters. The highest thruster array load observed during testing early in 1979 was 785 watts. This is just below the predicted peak power of 814 watts. Additional testing will provide data to further refine the present array characteristics, however, the final results are not likely to differ by more than a percent or two from the following findings: (see fig. 11)

Degradation of open circuit voltage	11.6%
Degradation of short circuit current	19.5%
Degradation of maximum power voltage	16.8%
Degradation of maximum power current	22.9%
Degradation of maximum power	35.9%

Degradation of the maximum power point is more than would be expected from the open-circuit voltage and short-circuit current degradation. This is because the "fill factor" or squareness of the I-V curve has decreased for the degraded array.

The predicted power degradation was 40.5 percent for early in 1979. See figure 6 for a plot of the projected radiation degradation. Less degradation than expected indicates no significant contamination has occurred from spacecraft sources including ion thruster operation.

Solar array temperatures in 1979 are comparable to those observed in 1970, indicating the solar array optical and thermal properties have remained fairly constant. Six of the 90 solar panels carry temperature instrumentation. These panels are located symmetrically with three on each wing (see fig. 9). The temperature gradient observed along a solar array wing has been factored into the array characteristics presented. Typically, the inboard edge of the solar array wing is warmer than the outboard edge with the midsection temperature a little warmer than the average of the edge temperatures. The magnitude of the temperature gradient (edge to edge) is a function of sun angle but is typically 6° C (11° F) to 9° C (16° F). The effect of the nonuniform array temperature is that the cooler panels set the array open-circuit voltage. As power is drawn from the array the cooler panels (higher voltage panels) provide current first. This results in a power mismatch due to unequal load sharing, that is, all panels do not reach their peak power operating point simultaneously. The degradation data presented accounts for the effects of the nonuniform array temperatures. As an example, the observed solar array open-circuit voltage was used in calculations involving outboard panels only. For warmer inboard panels, the observed solar array open-circuit voltage was adjusted (lowered) to account for the known temperature gradient. The solar array electrical characteristics presented in figure 11 represent array parameters adjusted to a uniform temperature.

As noted earlier, the resistivity of the SERT II solar cell is nominally 2-ohm-cm. Many of the NASA solar arrays flown to date use 10 ohm-cm resistivity cells. However, with the higher power demands of future satellites use of the more efficient 2 ohm-cm (and lower) resistivity is planned. The baseline SEP array employs a 2 ohm-cm cell. With one exception (Skylab) the SERT II solar array is the largest array flown by NASA. The SERT II array data provides the future designer with a data point on how these cells perform over a long period (years) in space. Additional information on the development and early flight performance of the SERT II solar array is covered in reference 9.

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TABLE 1. - OPERATION OF SERT II THRUSTER SYSTEM TWO

Run	Total starts, number	Run start, day 1979	I_5, mA (±3)	V_5, v (±35)	Hours on	Total system, hours	H.V. recycles	I_9, mA (±4)	Gimbal angle, θ, degrees	Reason run stopped
1	219	22	85	3150	63	2074	1	86	8.2	(A)
2	220	28	190	2610	0	2074	1	---	7.7	Ground station command error
3(a)	221	31	85	3150	118	2192	0	82	7.7/7.0/6.8	No stoppage, beam turned to higher level
3(b)	---	36	200	3080	26	2218		195	6.2	(A)
4	222	40	83	3080	194	2412		86/87	6.2	(B)
5	223	59	85	3080	25	2437		80	10‡	(A)
6	224	64	---	----	0	2437		--	7.4	(B)
7	225	64	---	----	0	2437		---	7.4	(B)
8	226	66	83	3150	49	2486	1	80	6.0	Commanded turn off
9	227	71	88	3080	50	2536	0	76	5.7	(A)
10	228	74	85	3080	25	2561	0	76	5.5	Commanded turn off
11	229	78	---	----	0	2561	-	---	5.5	Grid short
12	230	93	85	3080	49	2610	0	80	7.1	Commanded turn off
Total					599	2610[(C)]				

‡Limit of gimbal position.

(A) Solar array undervoltage turned off power processor during H.V. recycle.

(B) Solar array undervoltage turned off power processor during command to higher beam current.

(C) Total hours on cathodes in space, 2881 hr; on ground, 91 hr.

TABLE 2. - PERFORMANCE OF FLIGHT THRUSTER 2

		Preheat							Propellant, no beam						
	Year	1970	1973	1974	1975	1976	1977	1979	1970	1973	1974	1975	1976	1977	1979
	Day	2/11	6/1	10/7	12/4	4/22	11/29	1/22	2/11	6/14	8/23	12/4	9/22	11/29	1/22
	Restart number	10	80	213	215	216	218	219	10	86	195	215	216	218	219
Main vaporizer heater	V2,v	0	0	0	0	0	0	0	[g]1.63	[a]1.49	1.85	1.85	(f)	1.75	1.63
	I2,a	0	0	0	0	0	0	0	[g]1.41	[a]1.32	1.70	1.80		1.65	1.58
Main cathode heater	V3,v	16.0	15.6	15.6	15.6	15.0	15.4	15.5	8.7	9.5	9.1	9.1		6.4	5.9
	I3,a	2.86	2.81	2.81	2.81	2.81	2.87	2.87	1.54	1.57	1.57	1.67		1.61	1.57
Main discharge	V4,v	>50	>50	>50	>50	>50	>50	>50	39.9	39.7	40.4	40.4		40.7	39.4
	I4,a	0	0	0	0	0	0	0	2.0	2.3	1.7	1.7		1.8	1.9
Beam voltage	V5,v	0	0	0	0	0	0	0	0	0	0	0		0	0
Beam current	I5,a	0	0	0	0	0	0	0	0	0	0	0		0	0
Accelerator grid	V6,v	0	0	0	0	0	0	0	0	0	0	0		0	0
	I6,mA	0	0	0	0	0	0	0	0	0	0	0		0	0
Neutralizer heater	V7,v	[g]7.7	8.8	8.6	10.4	8.2	8.4	8.0	[g]7.7	[a]10.4	8.4	8.8		8.4	7.5
	I7,a	[g]2.3	2.6	2.5	3.0	2.5	2.5	2.3	[g]2.3	[a]3.0	2.4	2.4		2.5	2.3
Neutralizer keeper	V8,v	28.5	28.5	27.8	38.0	28.5	28.5	28.5	28.5	[a]32.3	28.5	28.5		28.5	28.1
	I8,a	[d]0.226	[d]0.183	[d]0.191	0.179	0.189	0.197	0.197	[d]0.199	[a]0.175	[d]0.179	[d]0.171		0.181	0.179
Spacecraft voltage	v	-6	(f)	-3	(f)	(f)	(f)	-9	-9	(f)	-4	(f)		(f)	-9
Neutralizer emission	a	0	0	0	0	0	0	0	0	0	0	0		0	0
Main cathode keeper	V10,v	[d]416	[d]363	[d]371	[d]411	[d]363	[d]386	[d]363	12.3	9.9	10.8	11.0		10.5	11.0
	I10,a	0	0	0	0	0	0	0	[b]0.280	[b]0.273	[b]0.273	[b]0.271		[b]0.279	[b]0.272
Solar array	v	70	62	65	77	63	67	61	38	61	60	59	61	65	62

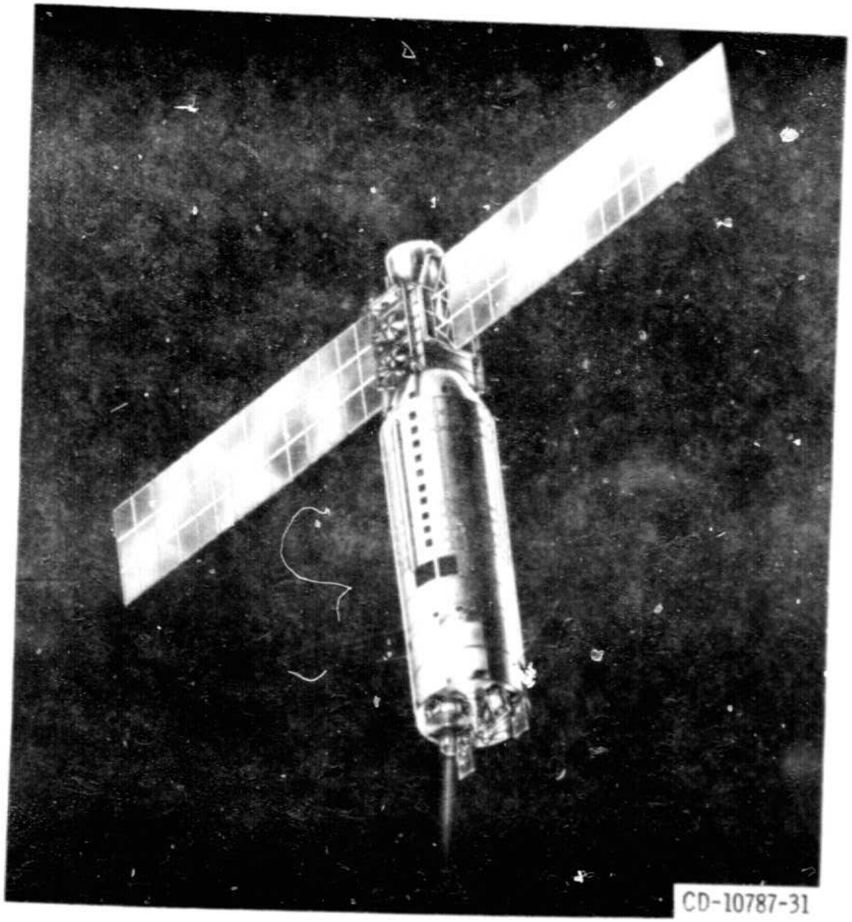
		30 percent beam								80 percent beam				Telemetry uncertainty (rms)
	Year	1970	1974	1975	1976	1977	1979	1979	1979	1970	1974	1975	1979	
	Day	2/11	9/10	12/4	9/22	11/29	1/22	1/22	4/5	2/11	9/11	12/4	2/5	
	Restart number	10	198	215	216	218	219	219	230	10	200	215	211	
Main vaporizer heater	V2,v	[g]1.63	1.70	1.70	[c]0.19	[a]2.67	1.70	1.70	1.70	1.70	1.85	1.85	1.85	±0.07
	I2,a	[g]1.51	1.77	1.70	[a]0.09	[a]2.97	1.67	1.67	1.67	1.70	1.95	1.96	1.85	±0.08
Main cathode heater	V3,v	7.9	8.7	8.7	(f)	(f)	8.2	8.2	8.2	8.3	8.7	8.2	8.2	±0.35
	I3,a	1.54	1.57	1.57	1.61	1.61	1.57	1.57	1.57	1.54	1.57	1.57	1.57	±0.05
Main discharge	V4,v	42.2	42.4	42.4	39.8	(f)	42.3	[c]38.3	42.3	41.5	41.4	41.5	41.4	±0.2
	I4,a	0.7	0.6	0.56	[a]2.47	(f)	0.60	0.56	0.60	1.2	1.1	1.1	1.2	±0.05
Beam voltage	V5,v	[d]3490	[d]2960	[d]2900	2500	2440	3030	3030	2960	[d]3160	[d]2630	[d]2500	[d]2630	±65
Beam current	I5,a	[d]0.088	[d]0.083	[d]0.083	[a]0.172	[d]0.119	0.088	0.083	0.083	[d]0.203	[d]0.198	[d]0.198	0.199	±0.005
Accelerator grid	V6,v	[d]-1730	[d]-1480	[d]-1430	[d]1100	(f)	-1530	-1530	-1480	[d]-1640	[d]-1330	[d]-1280	[d]-1340	±50
	I5,mA	1.1	0.9	0.8	0.5	(f)	1.1	1.1	1.1	1.4	1.4	1.3	1.5	±0.1
Neutralizer heater	V7,v	[g]6.6	8.1	7.7	7.1	7.1	7.5	7.5	7.5	[g]6.4	7.5	7.0	6.7	±0.25
	I7,a	[g]2.0	2.3	1.9	[a]2.6	[a]2.7	2.2	2.3	2.3	[g]1.9	2.2	2.1	2.0	±0.05
Neutralizer keeper	V8,v	27.8	27.8	27.8	28.5	29.2	28.5	[c]24.0	28.1	[c]24.0	[c]27.8	27.8	28.1	±0.7
	I8,a	[d]0.215	[d]0.175	[d]0.175	0.181	0.18	0.181	0.189	0.185	[d]0.206	[d]0.167	0.163	0.174	±0.004
Spacecraft voltage	v	-17	-8	(f)	(f)	(f)	-9	-9	-9	-17	(f)	(f)	-9	±2
Neutralizer emission	a	0.087	0.080	0.080	(f)	(f)	0.086	0.086	0.080	0.201	0.195	0.195	0.197	±0.006
Main cathode keeper	V10,v	20.4	20.0	20.0	13.8	15.7	18.9	18.9	19.8	13.9	13.1	13.1	12.9	±0.5
	V10,a	[b]0.272	[b]0.264	[b]0.260	[b]0.257	[b]0.256	[b]0.259	[b]0.258	[b]0.260	[b]0.272	[b]0.260	[b]0.259	[b]0.258	[b]±0.003
Solar array voltage	v	68	59	56	[h]50	[h]49	59	59	59	63	52	[h]50	53	±1.0

[a]Value changing in response to control signal.
[b]I10 value estimated from V10 value and power supply response characteristic curve.
[c]Values due to different set points.
[d]Difference in values due to different solar array voltages input to power processor.
[f]Data unavailable.
[g]Heater power lower due to higher thermal background.
[h]Value estimated from V5.

TABLE 3. - REPRESENTATIVE HEATER VALUES[c] AND CATHODE STARTING TIMES

Thruster	Start number	Date	Main vaporizer			Main cathode			Neutralizer cathode			Cathode start time		Total cathode, on time,[d] hr	Neutralizer reservoir temperature, °C
			I2, A	V2, V	V2/I2, Ω	I3, A	V3, V	V3/I3, Ω	I7, A	V7, V	V7/I7 Ω	Neutralizer cathode, min	Main cathode, min		
1	1	12/9/69	2.80	(a)	(a)	2.80	>15	>5.3	2.78	(a)	(a)	6.5	0.3	----	(a)
	4	12/28/69	2.81	(a)	(a)	2.92	15.7	5.4	2.79	9.9	3.6	6.2	.4	----	(a)
	5	2/14/70	2.81	2.74	0.98	2.88	15.7	5.5	2.90	10.3	3.6	3.3	.3	0	(a)
	6	3/8/70	2.89	(a)	(a)		15.3	5.3		10.6	3.7	4.2	.3	508	83
	7	5/21/70	(a)	2.67	(a)		15.3	5.3		10.8	3.7	4.3	.7	2283	78
	14	10/26/70	2.89	2.60	.90		14.1	4.9		10.8	3.7	4.2	[b]4.4	3794	47
	20	2/11/71	2.89	2.67	.93		15.7	5.5		10.3	3.6	4.2	.3	3855	83
	32	1/21/72	(a)	(a)	(a)		15.7	5.5	2.79	10.1	3.6	6.2	(a)	3868	29
	33	5/25/73	2.81	2.74	.97	2.82	15.3	5.4	2.90	10.6	3.7	[illegible]	[b]6.4	3869	(a)
	145	8/19/74	2.89		.95		15.3	5.4		10.8	3.7	6.3	7.4	3885	
	156	10/9/74	2.81		.98		15.7	5.6		10.3	3.6	6.9	9.5	3889	
	157	11/14/75	2.89		.95		15.3	5.4		10.6	3.7	6.4	3.8	3890	
	160	9/15/76	2.09	2.10	1.00		14.5	5.2				6.5	3.0	3891	
	167	12/14/76	2.70	2.74	1.01	2.88	15.7	5.5				5.8	7.2	3894	
	172	7/2[illegible]/77	2.79	2.74	.98	2.82	(a)	(a)				6.6	7.5	3896	
	176	11/22/77	2.89	2.74	.95	2.82	14.9	5.3				5.8	3.1	3897	
	177	7/14/78		2.67	.92	2.88		5.2		10.3	3.6	6.5	2.9	3898	
	178	7/14/78		2.74	.95	2.88		5.2		10.6	3.7	5.1	1.5	3898	
	179	1/17/79		2.67	.92	2.82		5.3		10.3	3.6	4.8	1.0	3904	
	185	3/1[illegible]/79		2.67	.92	2.82	14.5	5.2		10.3	3.6	4.8	4.0	4046	
	191	5/22/79		2.67	.92	2.88	14.9	5.2		10.3	3.6	6.6	6.7	4085	
2	1	11/29/69	2.89	(a)	(a)	2.78	>15	>5.4	2.94	(a)	(a)	10.0	1.0	----	(a)
	4	12/21/60	2.90	(a)	(a)	2.77	16.0	5.8	2.86	(a)	(a)	6.3	1.0	----	(a)
	10	2/11/70	2.88	2.77	.96	2.86	16.0	5.6	2.97	10.2	3.4	3.2	.4	0	97
	11	7/24/70	2.97	2.70	.91	2.86	16.0			10.2	3.4	3.2	.9	38	97
	12	9/2/70				2.81	15.6			10.4	3.5	3.7	.9	934	65
	53	11/13/70				2.81	15.6					2.8	.9	2094	69
	67	2/26/71				2.86	16.0					2.7	.4	2126	115
	76	1/21/72	(a)	(a)	(a)	2.86	16.0					5.3	(a)	2149	33
	126	7/17/73	2.97	2.70	.91	2.81	16.0	5.7				5.2	[b]8.2	2162	22
	189	8/19/74					15.6	5.6				5.4	10.5	2166	43
	203	9/12/74								10.2	3.4	6.1	22.5	2169	40
	211	10/2/74								10.2	3.4	6.8	12.7	2175	35
	215	12/4/75								10.4	3.5	7.4	29.3	2177	37
	216	9/22/76				2.86	15.0	5.3	2.92	10.2	3.5	6.2	[b]41.0	2178	30
	218	11/29/77					15.4	5.4		10.0	3.4	6.4	2.1	2279	33
	219	1/22/79					15.6	5.3		10.2	3.5	3.9	2.0	2342	62
	226	3/7/79					14.8	5.0		10.4	3.6	3.8	1.8	2757	62
	230	4/3/79		2.63	.69		14.1	4.8		10.2	3.5	3.2	6.7	2881	97

[a]Data not taken or unavailable.
[b]No preheat used.
[c]Quantizing and calibration error, ±3%, root-sum-square.
[d]Includes heating time in space only; ground time, thruster 1 - 83 hr, thruster 2 - 91 hr.



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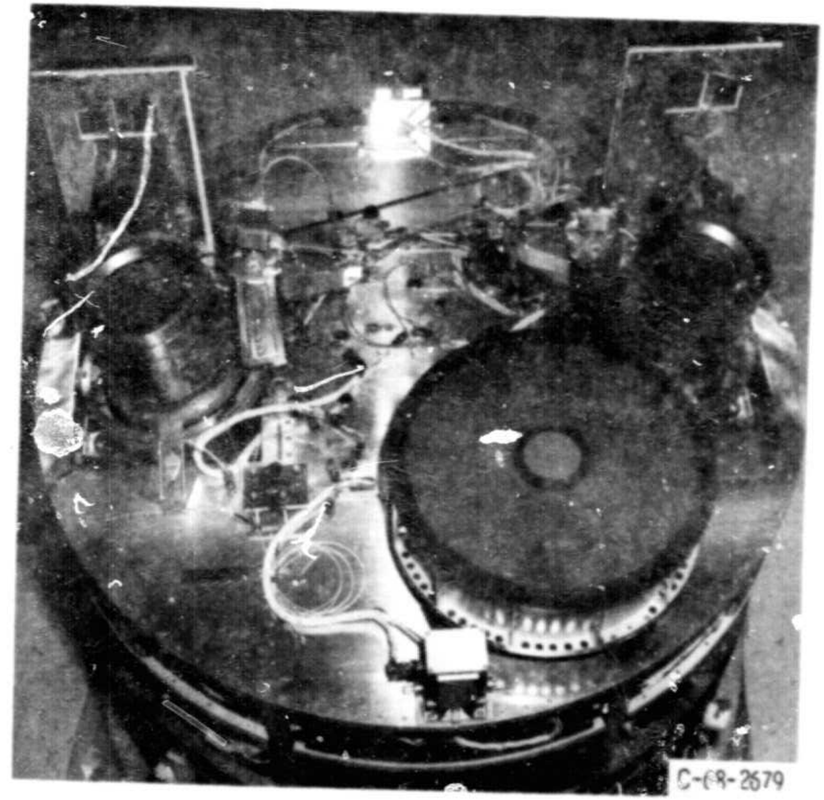
(a) SPACECRAFT IN ORBIT, (ARTIST'S CONCEPTION).

Figure 1. - SERT II spacecraft.



C-68-2679

(b) THRUSTER END OF SPACECRAFT.

Figure 1. - Concluded.

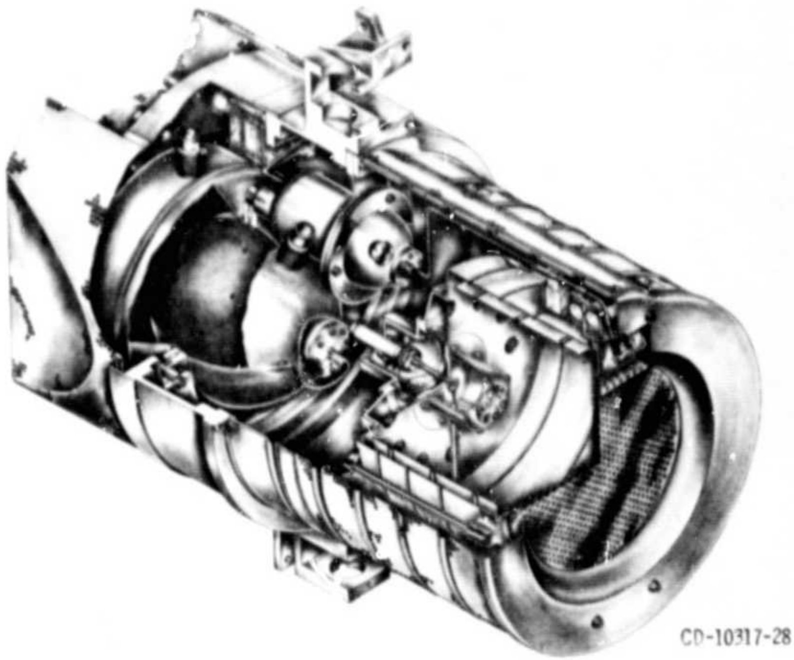


Figure 2. - SERT II Ion Thruster.

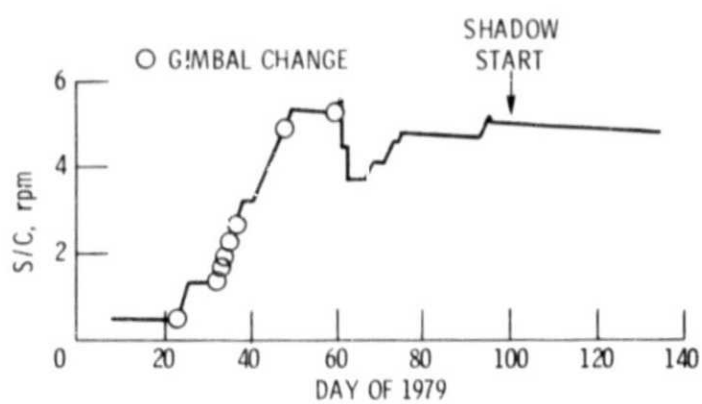


Figure 3. - SERT II spacecraft spin rate.

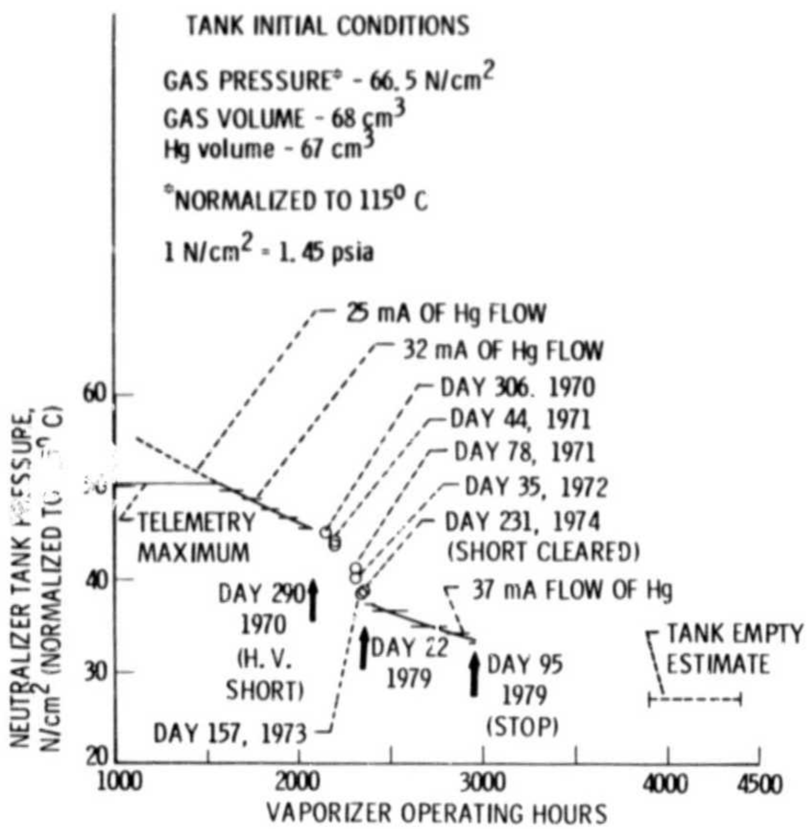


Figure 4. - SERT II neutralizer-2 tank pressure log.

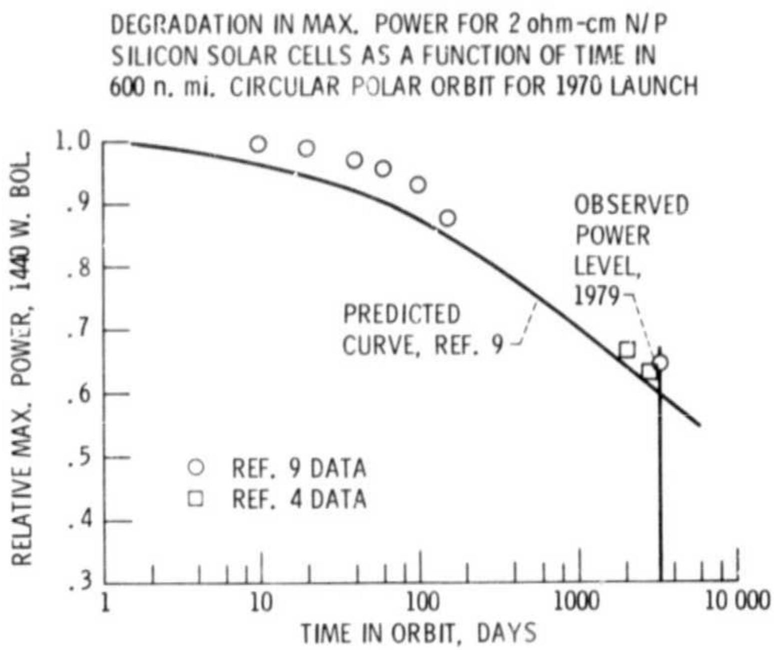


Figure 5. - SERT II solar array radiation degradation.

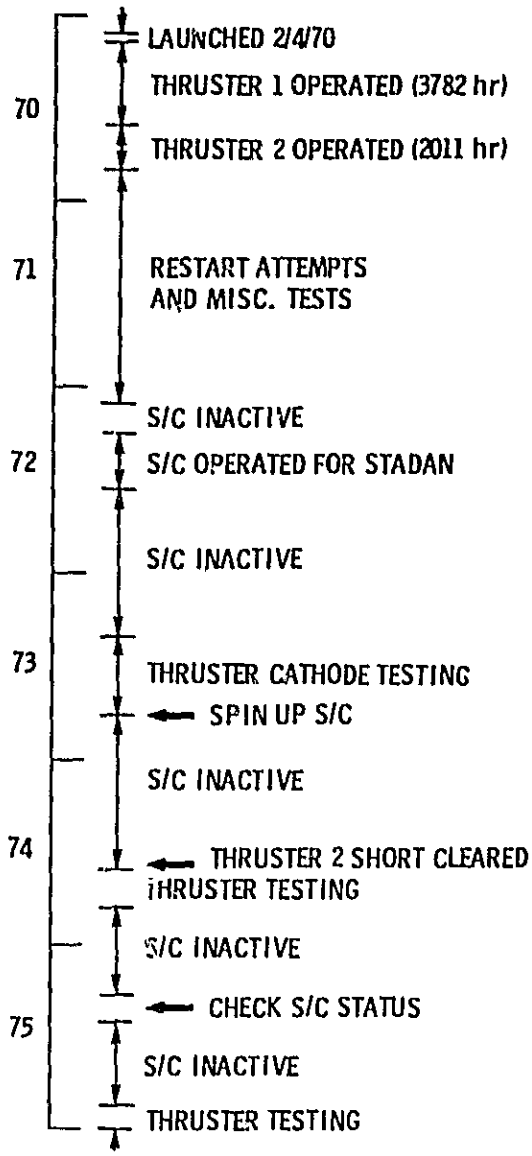


Figure 6. - SERT II operational history.

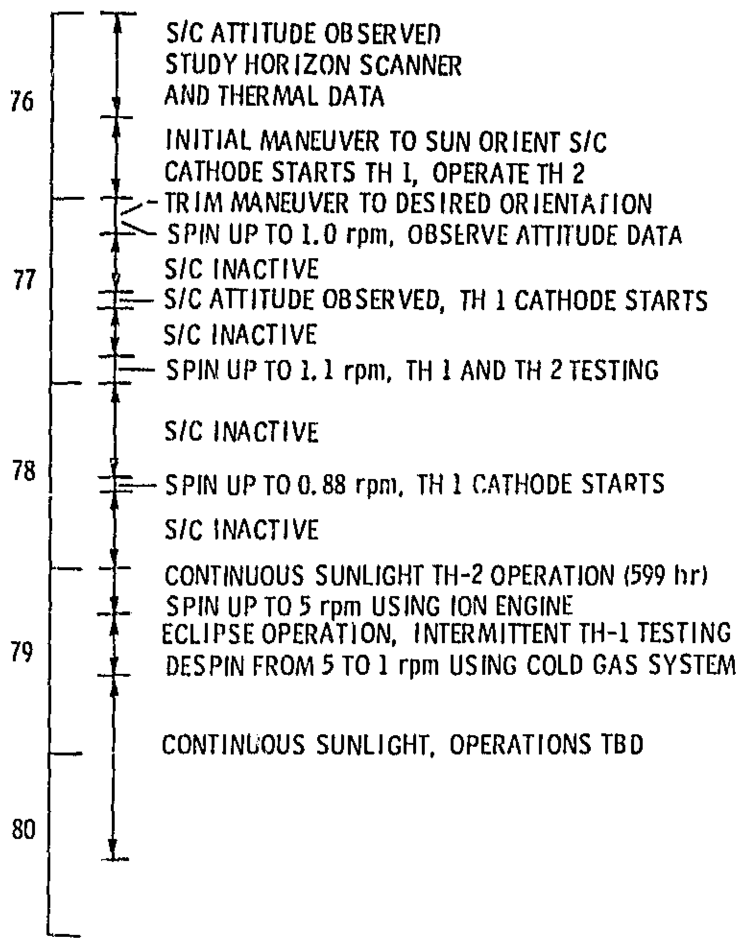


Figure 6. - Concluded.

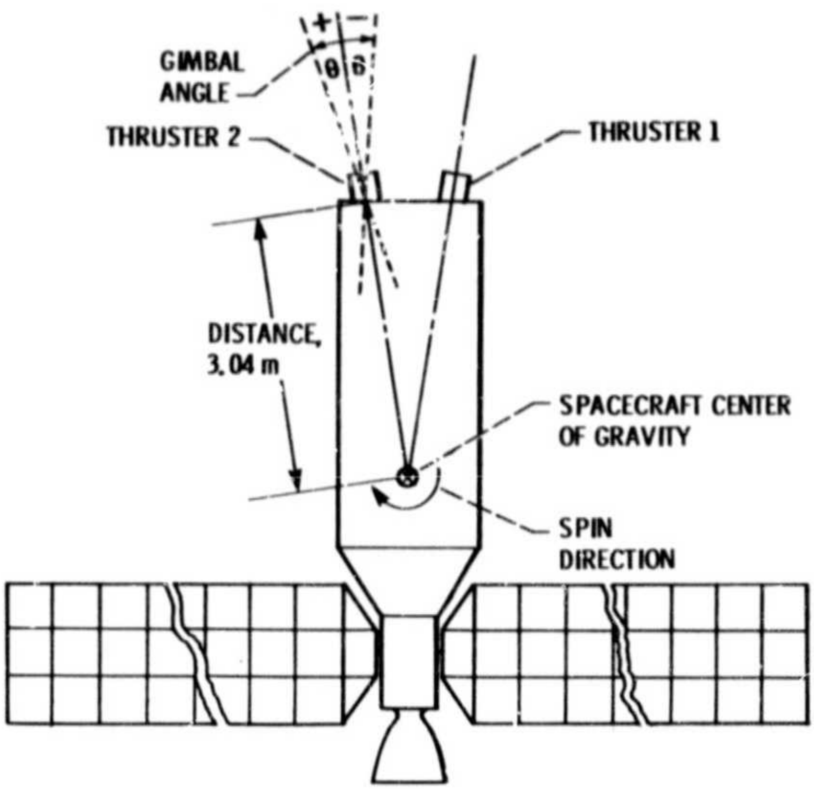
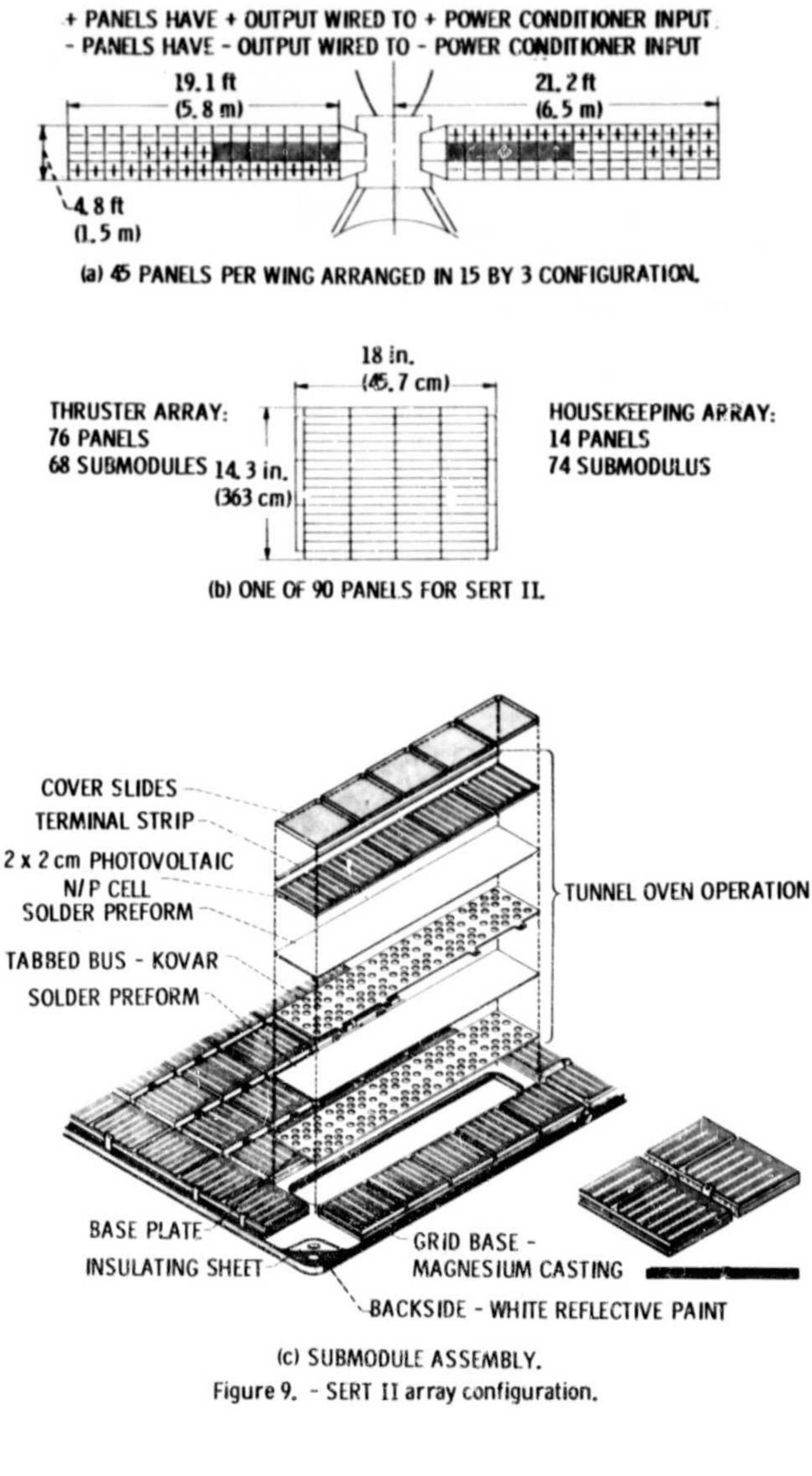


Figure 7. - SERT II spacecraft showing ion thruster gimbal angle.

Figure 8. - SERT II Ion thruster gimbal structure.



(a) 45 PANELS PER WING ARRANGED IN 15 BY 3 CONFIGURATION.

(b) ONE OF 90 PANELS FOR SERT II.

(c) SUBMODULE ASSEMBLY.

Figure 9. - SERT II array configuration.

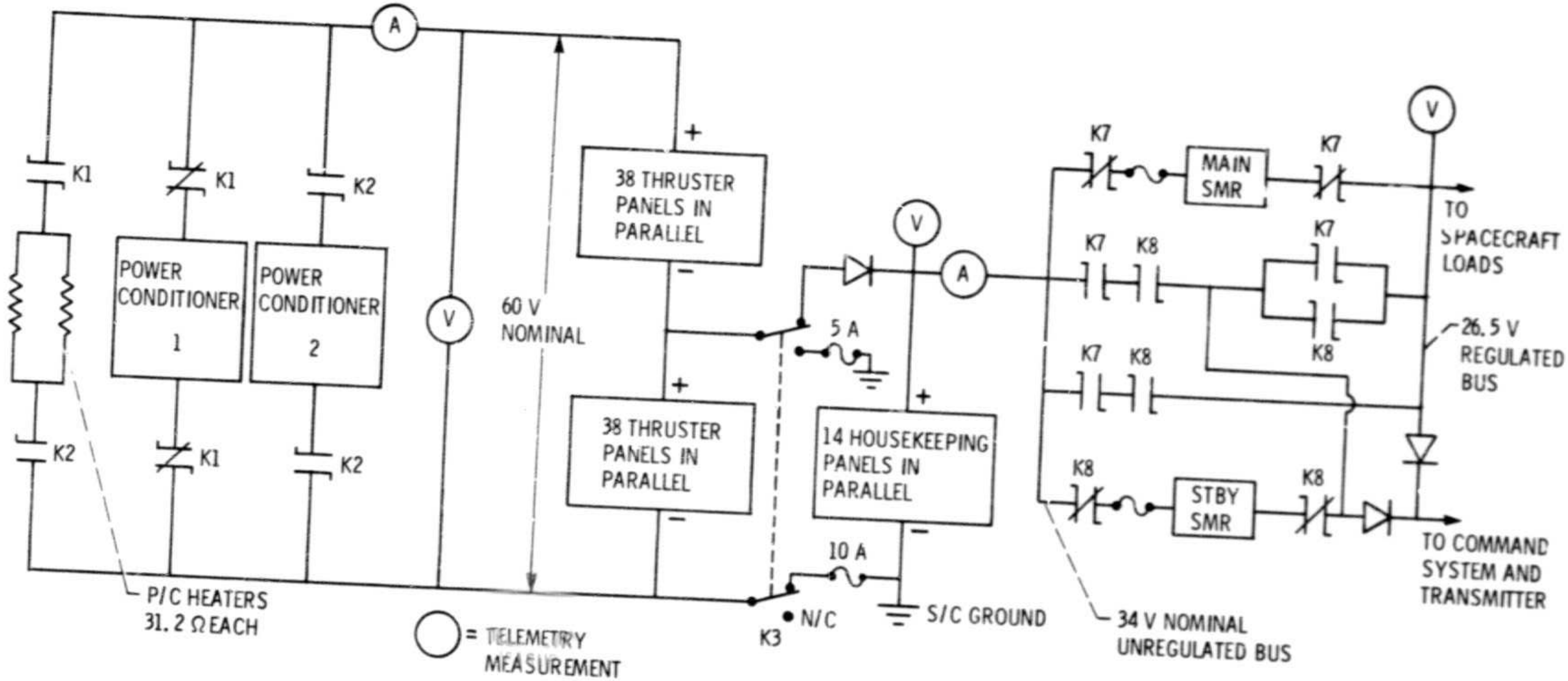


Figure 10. - Simplified diagram SERT II power system.

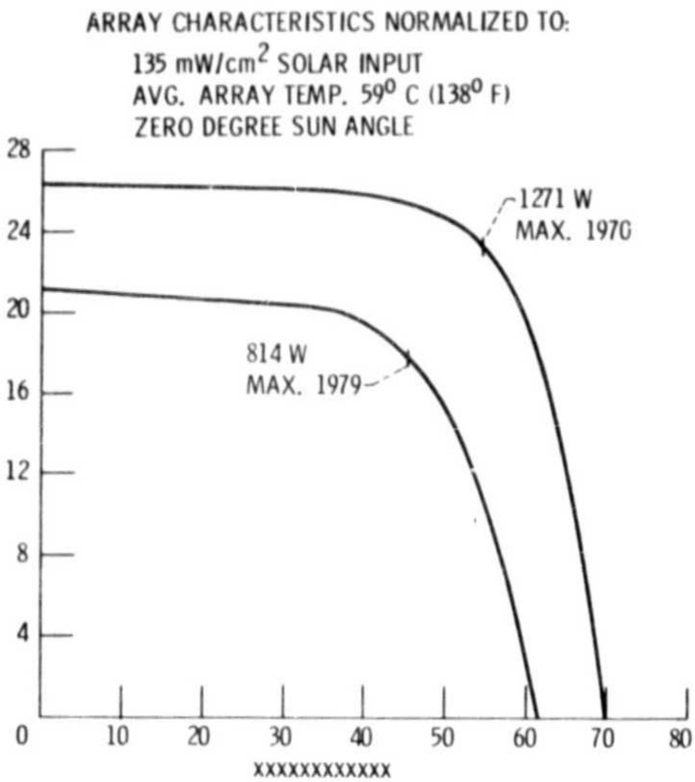


Figure 11. - SERT II solar array IV curves 1970, 1979.